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HIGH/VARIABLE MIXTURE RATIO O2/H2 ENGINE

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ABSTRACT

Vehicle/engine analysis studies have identified the High/Dual Mixture Ratio O2/H2 Engine cycle as a leading candidate for an advanced Single Stage to Orbit (SSTO) propulsion system. This cycle is designed to allow operation at a higher than normal O/F ratio of 12 during liftoff and then transition to a more optimum O/F ratio of 6 at altitude. While operation at high mixture ratios lowers specific impulse, the resultant high propellant bulk density and high power density combine to minimize the influence of atmospheric drag and low altitude gravitational forces. Transition to a lower mixture ratio at altitude then provides improved specific impulse relative to a single mixture ratio engine that must select a mixture ratio that is balanced for both low and high altitude operation. This combination of increased altitude specific impulse and high propellant bulk density more than offsets the compromised low altitude performance and results in an overall mission benefit.

This paper will address two areas of technical concern relative to the execution of this dual mixture ratio cycle concept. First, actions required to transition from high to low mixture ratio are examined, including an assessment of the main chamber environment as the main chamber mixture ratio passes through stoichiometric. Secondly, two approaches to meet a requirement for high turbine power at the high mixture ratio condition are examined. One approach uses high turbine temperature to produce the power and requires cooled turbines. The other approach incorporates an oxidizer-rich preburner to increase turbine work capability via increased turbine mass flow.

INTRODUCTION

A reusable SSTO is potentially the lowest cost system for Earth to LEO operations with small to medium size payload (approximately 50K lbs), but is extremely sensitive to vehicle mass fraction and propulsion system performance. For SSTO, as with any rocket system, altitude matched performance is extremely important. A two stage vehicle can optimize booster performance for low altitude operation and upper stage performance for high altitude operation almost independently. For a SSTO vehicle, the same propulsion system must meet both sets of performance requirements. During the initial portion of flight, engine thrust to weight ratio and high bulk density propellants are the critical operating parameters with specific impulse being of lesser importance. For latter portions of the ascent, specific impulse becomes the dominating parameter, with adequate throttling capability to limit maximum dynamic pressure and maximum G forçes also important.

The dual mixture ratio propulsion concept is designed to allow operation at low nozzle area ratio and high O/F ratio during the initial phase of the flight. The low area ratio nozzle and high oxidizer mass flow rate provide the higher engine thrust to weight characteristic. The increased use of oxygen provides the high vehicle bulk density characteristic. At altitude, the nozzle area ratio is increased via a deployable nozzle extension and the engine transitions to a more optimum O/F ratio of 6 in order to maximize specific impulse. Figure 1 is a schematic of the

dual mixture ratio cycle, with critical operating parameters identified for both high and low mixture ratio modes. Earth to orbit engine/vehicle studies indicate that up to a 15% reduction in vehicle dry weight is possible with this dual mixture ratio concept, as compared to a conventional staged combustion configuration.

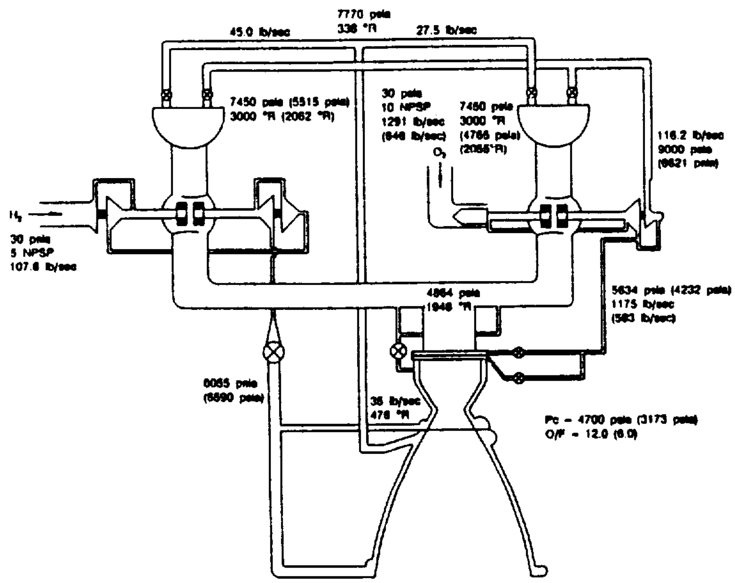

Figure 1. Dual Mixture Ratio Staged Combustion Engine Cycle

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MIXTURE RATIO TRANSITION INVESTIGATION

The dual mixture ratio transition is accomplished by holding total engine fuel flow constant while reducing the oxidizer flow by 50%. Table 1 is a summary of the normalized valve areas for the 6 and 12 mixture ratio steady state conditions. As shown, the transition is accomplished by resetting only three valve areas; the main oxidizer valve area (MOVA), the oxidizer preburner oxidizer valve area (OPOVA), and the oxidizer preburner fuel valve area (OPFVA). All other valve areas can remain constant. A quasi steady-state valve sequence study was performed to gain insight into propulsion system response to various valve closing sequences. The results of this investigation indicate that the oxidizer preburner must be throttled first, with OPFVA and OPOVA scheduled together to avoid unnecessary turbine thermal cycling. Adequate fuel and oxidizer turbopump spooldown time must be provided before MOVA throttling can be accomplished without overspeeding the turbomachinery. This MOVA throttling action must be timed in order to control turbine system back pressure and to achieve a smooth thrust transition.

After examination of the overall system quasi steady-state behavior, it is estimated that this transition can be accomplished in less than one second.

TABLE-1 Normalized Valve Area Summary

O/F	MAIN 02 (MOVA)	FUEL PREBURNER 02-(FPOVA)	FUEL PREBURNER H2-(FPFVA)	OXIDIZER PREBURNER 02-(OPOVA)	OXIDIZER PREBURNER H2-(OPFVA)
12.	1.0	1.0	1.0	1.0	1.0
6.	0.5	1.0	1.0	0.3	0.1

Another area of concern is that as the mixture ratio transitions from an O/F of 12 to 6, the main combustion chamber and nozzle are exposed to stoichiometric conditions (O/F = 8). Although the control valve study indicates that the transition time will be short, there is a chamber cooling concern to be addressed. Figures 2 and 3 show estimates of main chamber pressure and combustion efficiency as a function of main chamber O/F ratio. Since the main combustion chamber has been optimized for performance at O/F=6, there is a noticeable falloff in combustion efficiency at O/F=12. A preliminary thermal analysis was performed using these pressure and efficiency trends to generate a 'worst case' steady state heat flux estimate at the nozzle throat. The results of this study, shown in Figure 4, indicate at most a 20% increase in heat flux during this brief transition. This heat flux is only about 20% above levels currently encountered during operation of the SSME and is considered well within the range of technology improvements that could be made available for SSTO.

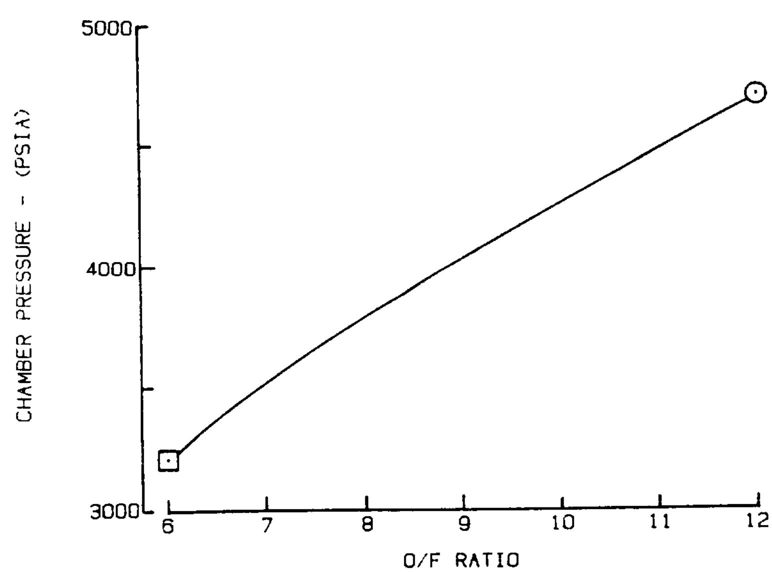

Figure 2. Main Chamber Pressure

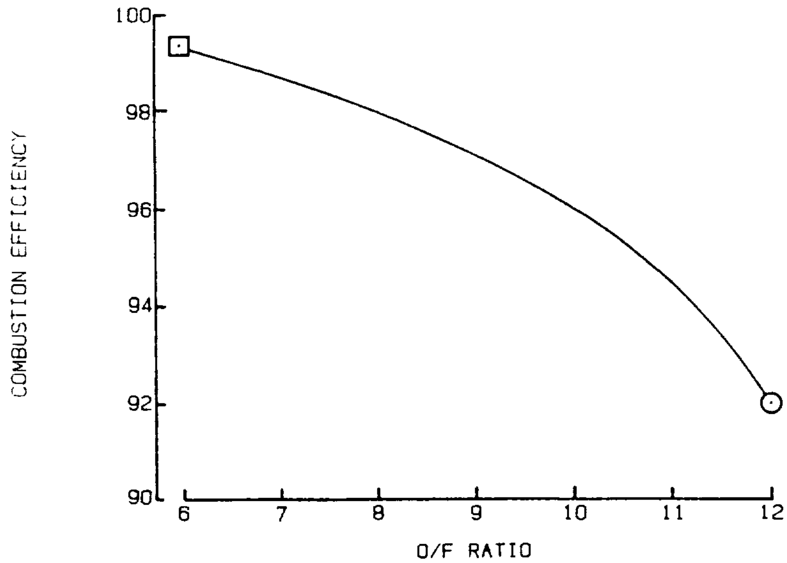

Figure 3. Main Chamber Combustion Efficiency

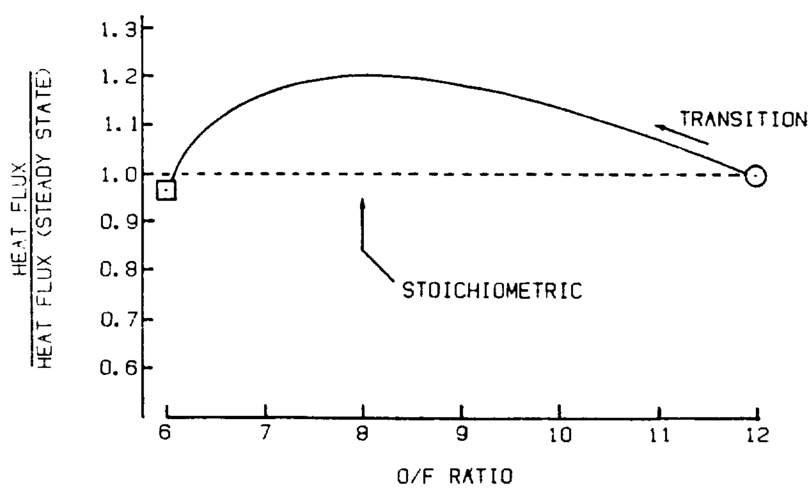

Figure 4. Main Chamber Heat Flux

TECHNOLOGIES FOR INCREASED TURBINE POWER REQUIREMENTS

In order to obtain maximum benefit from the dual mixture ratio concept, increased chamber pressures are required while operating in the high mixture ratio mode. This results in preburner turbine temperatures that exceed current uncooled material capabilities. Two possible solutions to this temperature concern are those of an oxidizer-rich preburner cycle or the use of

actively cooled turbines. Both concepts have potential to allow the staged combustion cycle to operate efficiently at the very high chamber pressures required while operating in the high mixture ratio mode. However, each concept will require additional enabling technology development to allow its successful incorporation into an advanced SSTO.

Oxidizer Rich Preburner

Typically, a staged combustion cycle is configured such that the preburner oxidizer flow is limited to the minimum required to sustain the desired preburner combustion temperature, with the remainder of the oxidizer flow being injected directly into the main chamber. This discussion will address the potential benefits and pitfalls that might Be encountered if the main chamber oxidizer flow stream were increased in pressure sufficiently to be injected into the preburner. The increased preburner mass flow, made available to the turbines via this concept, could then be used to reduce turbine inlet temperature and/or reduce turbine expansion ratio (i.e. reduce maximum overall system pressure for the same chamber pressure.)

Design point performance studies were conducted for the dual mixture ratio SSTO cycle with both fuel-rich and oxidizer-rich preburner cycles. The oxidizer-rich cycle was structured to use the increased turbine mass flow to reduce operating temperatures as opposed to system pressures. This option was selected because the baseline (fuel-rich) system pressures are in line with current technology but the turbine temperatures would most likely require active cooling. Although extra oxidizer turbine work is required to increase the oxidizer flow to preburner pressure levels, the increased mass flow allows the cycle to rematch with fuel and oxidizer turbine temperatures reduced by 500 and 1100 deg R ,respectively, from the 3000 deg R levels required by the fuel-rich cycle. It should be noted that only one preburner can be operated oxidizer-rich, as this is the only method to extract extra turbine work from the oxidizer flow with a net cycle benefit.

Although, the thermodynamic advantages of the oxidizer-rich preburner cycle could allow the use of uncooled turbines, significant hardware complications are expected to accompany the execution of this concept. Of primary concern is the problem of mixing the hot fuel-rich and oxidizer-rich preburner streams prior to combustion in the main chamber. Also of concern is the prolonged exposure of the highly stressed turbine hardware to the high temperature, high pressure, oxygen-rich environment.

Figure 5 illustrates the complexity that may be required in order to achieve controlled mixing of the fuel-rich and oxidizer-rich preburner discharge streams. The swirling mixer (swixer) must mix the streams with minimum pressure loss and set-up stable, efficient combustion in the main chamber. Because of the high energy of the preburner discharge streams, any premature mixing through a leak in the swixer could result in catastrophic distress of the hot gas manifold and main chamber. To further complicate the problem, the swixer, will most probably have to be cooled.

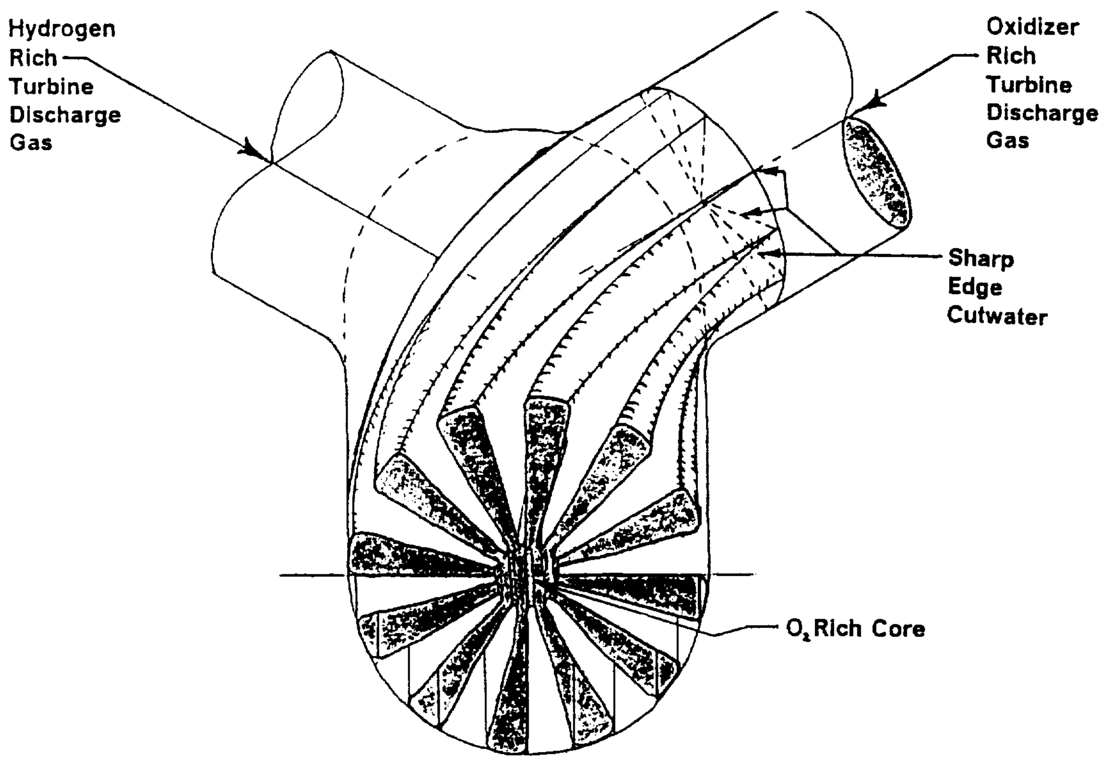

Figure 5. Conceptual Oxidizer Rich & Hydrogen Rich Swixer/Burner

The effects of exposure of the rotating turbine hardware to the high pressure and high temperature environment of the oxidizer-rich flow stream are not well known. The argument can be made that gas turbine engine turbines are exposed to similar conditions daily. One must consider, however, that typical operating pressures of jet engines seldom exceed 500 psia and to achieve the high operating temperatures, much of the oxygen is used in the combustion process. By conservative estimate, 'oxidizer-rich' turbine hardware for SSTO would be exposed to partial pressures of oxygen 100 times greater than a typical gas turbine engine at temperatures and stress levels that would be just as severe. A massive materials program would be required to characterize the oxidation and self-sustained combustion properties of materials that may be exposed to these demanding conditions. It would also be expected that most design procedures and guidelines would have to be reevaluated to assure successful designs in this type of environment.

Cooled Rocket Turbines

The idea of cooling turbine airfoils is not new. High temperature cooled turbines can be identified as the major contributor to most of the significant gas turbine engine performance advances over the past decade. Advanced cooling techniques and advanced materials have allowed these turbine temperature increases with enough design margin to achieve over a ten fold increase in turbine life. This high payoff for gas turbines has justified the continued private and

military funded programs that have advanced these technologies tremendously. It is felt that a major portion of these technologies can be applied directly to rocket turbines.

The point should be made, however, that the physics of cooling a rocket airfoil is very much different from cooling a gas turbine engine airfoil. At the high engine mixture ratio condition, the SSTO engine will operate at pressures above 8000 psia, which are approximately 20 times the levels run in gas turbine engine turbines. These pressure levels combined, with the fluid properties of hydrogen, will make the job of cooling much more difficult than a gas turbine engine. For instance, conventional gas turbine turbine cooling schemes rely significantly on convection cooling, whether it's impingement, radial flow or flow through or across turbulators. This convective concept will not work in the SSTO environment using nickel base alloys due to the high pressure rocket environment which creates very high heat flux through the airfoil wall. These heat flux levels result in temperature gradients which would not only crack the nickel wall but would provide very little cooling to the outer surface. Other materials with very high conductivity are more compatible with convective cooling techniques. The SSME combustor liner for example, is convectively cooled, but uses a highly conductive copper alloy to minimize thermal gradients. Unfortunately, the poorer strength of these alloys essentially prohibit their use in the high stress rotating turbine environment.

It is felt that other cooling techniques such as film cooling are better suited to the demanding rocket turbine environment. Film cooling techniques essentially insulate the airfoil with a very thin layer of coolant flow which must be distributed over the outside surface of the airfoil. The cooling hole pattern and film cooling hole shapes are tailored for each particular application. This concept has been studied for rocket turbines at Pratt and Whitney under contract NAS8-33821 and the results are summarized in Ref 1. Although this cooling method can be effective, relatively large quantities of coolant flow are still required to film cool the entire outside surface of the airfoil. It is estimated that each airfoil row will require approximately 6% of the turbine mainstream flow to cool it, based upon analysis of film cooling for rocket turbines presented in Ref. 1 and Ref. 2.

Past studies of cooled turbine rocket cycles, with cooling flow levels similar to the 6% per airfoil row , have shown little overall system payoff. It is not felt that cooling technology has advanced to such a level as to significantly change that result. What has changed, however, is the proposed turbine configuration to be cooled. A medium risk turbopump configuration philosophy has been conceived for advanced SSTO application. The turbopumps are each configured with two counter rotating spools that will allow a significant reduction in the total number of turbine airfoil rows and the elimination of all stationary turbine vanes. A conceptual representation of this concept, as it would be incorporated into an advanced SSTO fuel and oxidizer pump configuration, is shown in Figure 6. Figure 7 shows the type of turbine airfoil shapes typical of this counter-rotating, vaneless turbine concept. Also shown is a preliminary film cooling concept for the airfoils. It is felt that development of this airfoil efficient aerodynamic technology, combined with advanced film cooling techniques will be required to make cooled rocket turbines a viable option.

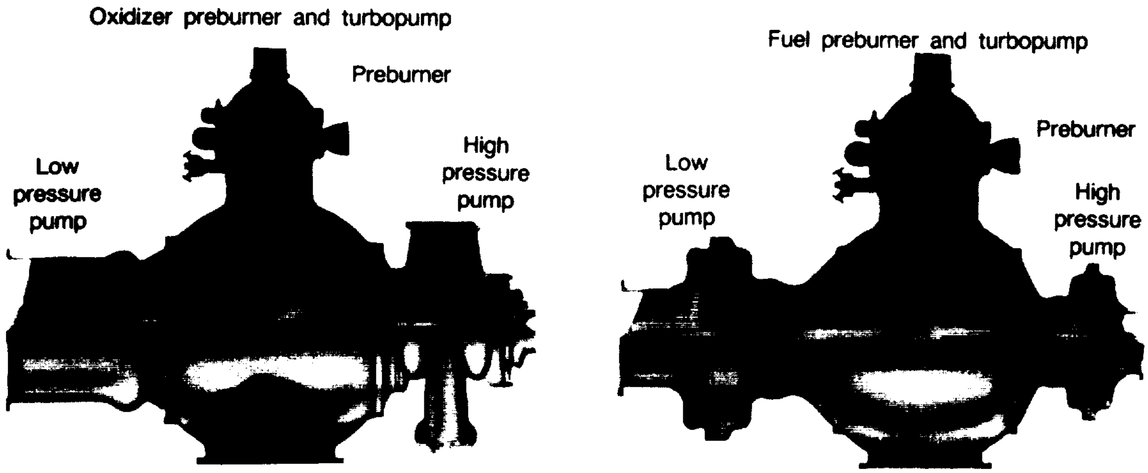

Figure 6. Conceptual SSTO Turbopump Configurations

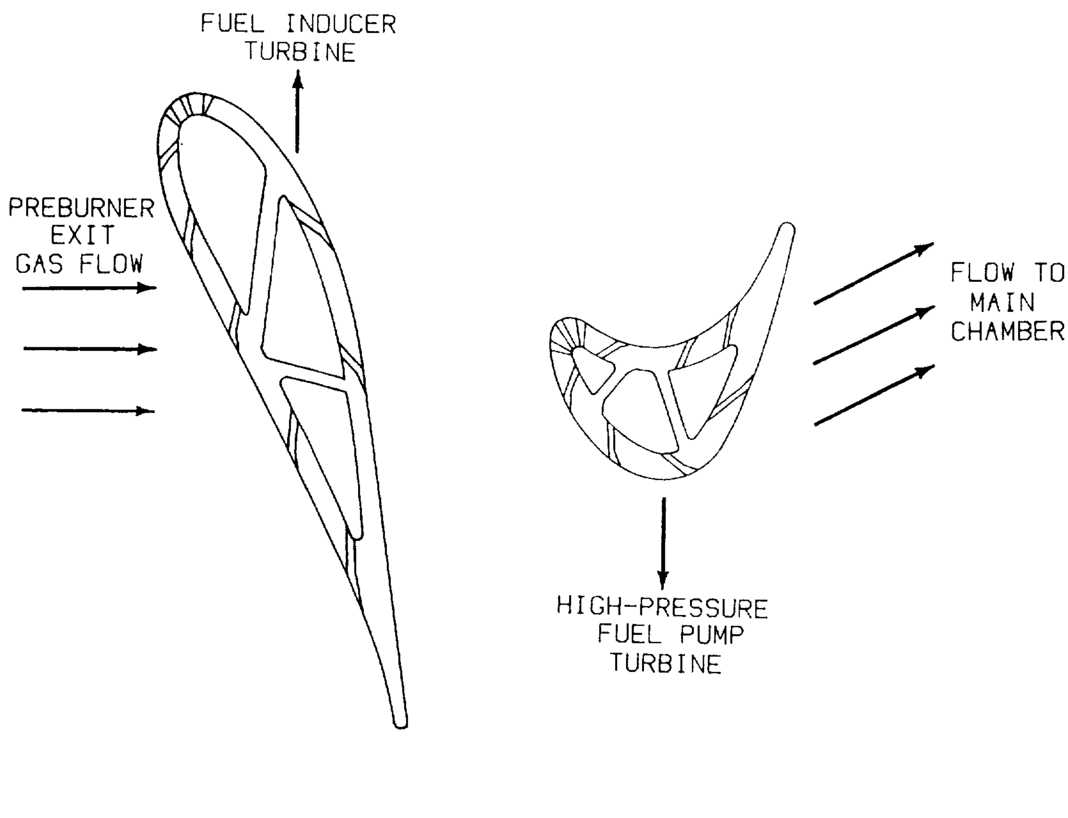

Figure 7. 'Vaneless' Airfoil Meanline Sections

SUMMARY

Development of the dual mixture ratio cycle concept will provide the capability to more optimally tailor the propulsion system performance for both low and high altitude operation. The resulting mission benefits can provide up to a 15% dry vehicle weight advantage, for an advanced SSTO application, relative to a conventional staged combustion cycle.

The capability to transition from the low altitude to the high altitude mode of operation will require only moderate advances in technology. Results of the investigation into the control philosophy to be required by this cycle, indicate that the mixture ratio transition can be accomplished in less than one second. Potential thermal problems that may be encountered as the main chamber transitions through stoichiometric, have been examined. The estimated 20% increase in heat flux that could be encountered, relative to current engines, will require additional cooling technology development but is not considered a major barrier.

Of the two technologies options that were examined as candidates to address the high turbine work level requirements for an advanced SSTO application, the results indicate that incorporation of cooled turbines, is the preferred technology path to pursue. Of primary concern with the oxidizer-rich concept is the controlled mixing of a the hot fuel-rich and the hot oxidizer-rich preburner discharge streams upstream of the main combustion chamber and the exposure of highly stressed turbine hardware to the harsh oxygen-rich environment. The recommended approach of actively cooled turbines, however, depends on the successful development of counter-rotating vaneless turbine aerodynamics.

REFERENCES

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2. High-Temperature Turbine Study, Rocketdyne Report, August 1986, Contract F04611-81-C-0053 with AFRPL.